MSFC-397-Abstract

Abstract

Solar Electric & Chemical Propulsion Technology Applications to a Titan Orbiter/Lander Mission

Several advanced propulsion technology options were assessed for a conceptual Titan Orbiter/Lander mission. For convenience of presentation, the mission was broken into two phases: interplanetary and Titan capture. The interplanetary phase of the mission was evaluated for an advanced Solar Electric Propulsion System (SEPS), while the Titan capture phase was evaluated for state-of-art chemical propulsion (NTO/Hydrazine), three advanced chemical propulsion options (LOX/Hydrazine, Fluorine/Hydrazine, high Isp mono-propellant), and advanced tank technologies. Hence, this study was referred to as a SEPS/Chemical based option. The SEPS/Chemical study results were briefly compared to a 2002 NASA study that included two general propulsion options for the same conceptual mission: an all propulsive based mission and a SEPS/Aerocapture based mission. The SEP/Chemical study assumed identical science payload as the 2002 NASA study science payload. The SEPS/Chemical study results indicated that the Titan mission was feasible for a medium launch vehicle, an interplanetary transfer time of approximately 8 years, an advanced SEPS (30 kW), and current chemical engine technology (yet with advanced tanks) for the Titan capture . The 2002 NASA study showed the feasibility of the mission based on a somewhat smaller medium launch vehicle, an interplanetary transfer time of approximately 5.9 years, an advanced SEPS (24 kW), and advanced Aerocapture based propulsion technology for the Titan capture. Further comparisons and study results were presented for the advanced chemical and advanced tank technologies.

MSFC-397-Presentati

Solar Electric & Chemical Propulsion Technology Applications to a Titan Orbiter/Lander Mission

2007 Joint Propulsion Conference

Michael Cupples

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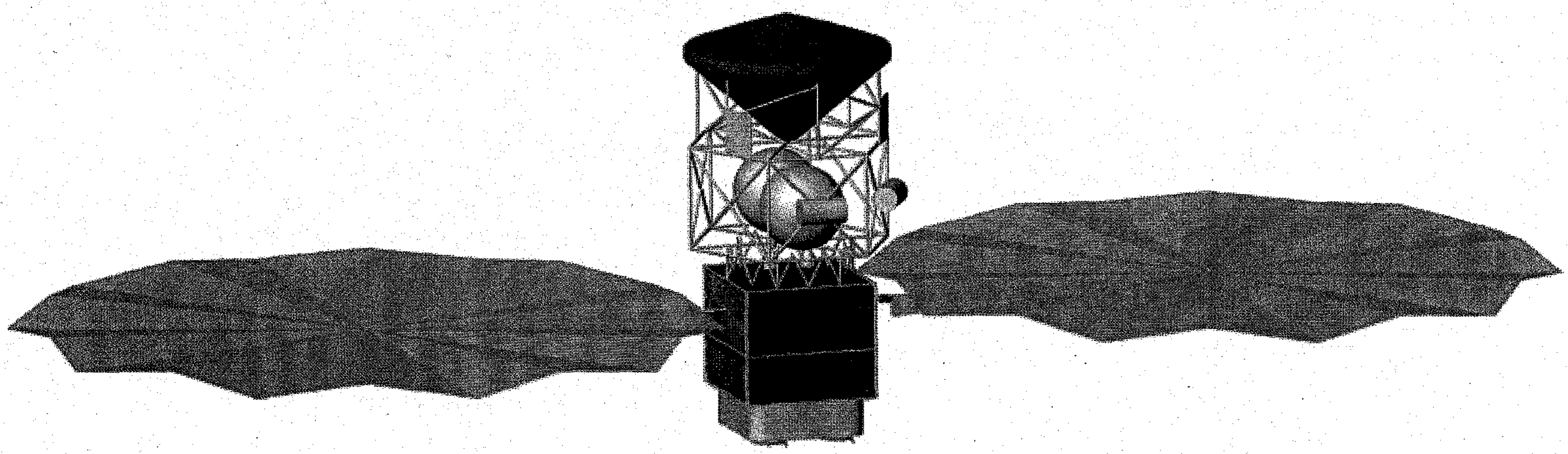


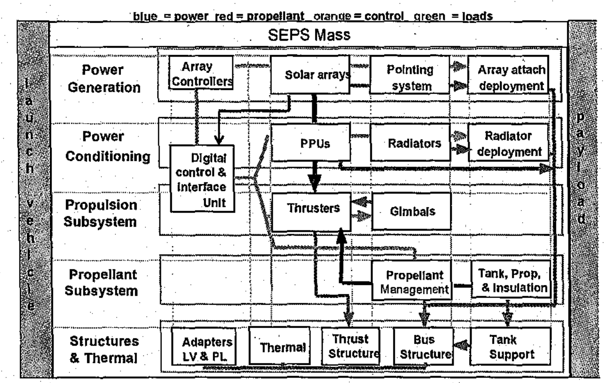
Objectives

- Define the performance of various propulsion options for a conceptual Titan mission
- Propulsion options investigated include
  - NEXT SEPS with chemical propulsive Titan capture
- Compare analysis results with 2002 NASA ISPT study results*
  - NEXT SEPS with Aerocapture at Titan
  - All-propulsive

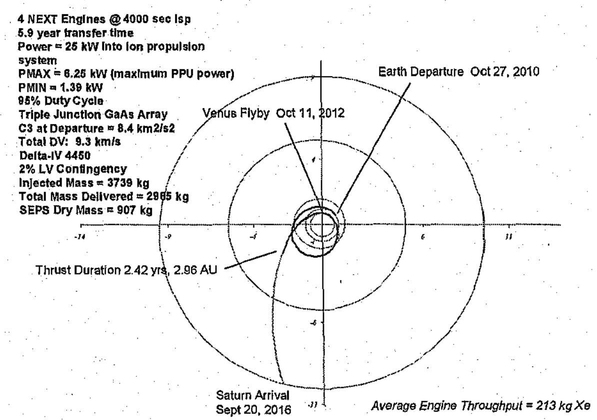
* Robert Bailey, ISPT Aerocapture Systems Analysis Review, August 2002

Overall Analyses Approach

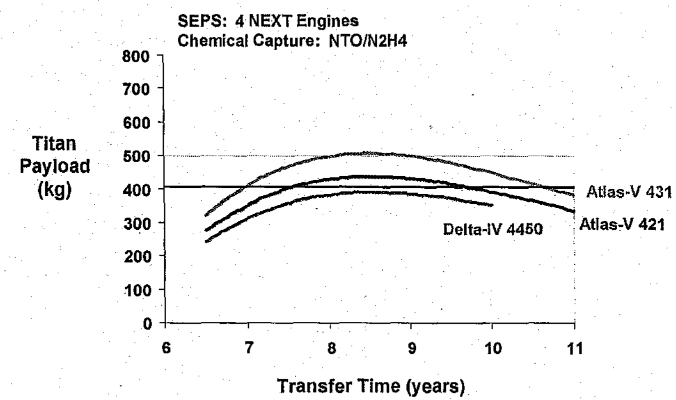
- **Systems Analysis**
  - **System models**
    - **Launch Vehicles**
    - **SEPS**
    - **Chemical Orbiter**



- **Mission Analysis**
  - **Trajectory optimization**
    - **SEPTOP Optimization Code**
    - **Launch Vehicle Models**
    - **Ga/As Array Model**
    - **Multi-Thruster NGI Models**



- **Basic figures of merit**
  - **Titan payload**
  - **Transfer time**
  - **Launch vehicle**



Global Key Analysis Assumptions[1]

Destination: Titan

Launch vehicles considered: Delta Medium, Delta Heavy, Atlas Medium

Final Titan orbit: Circular 2000 km altitude (NASA ISPT study case choose 1700 km for Titan science orbit)

Reference Payloads: Orbiter[2] 42 kg Science payload + Lander[3] 364 kg = 406 kg total

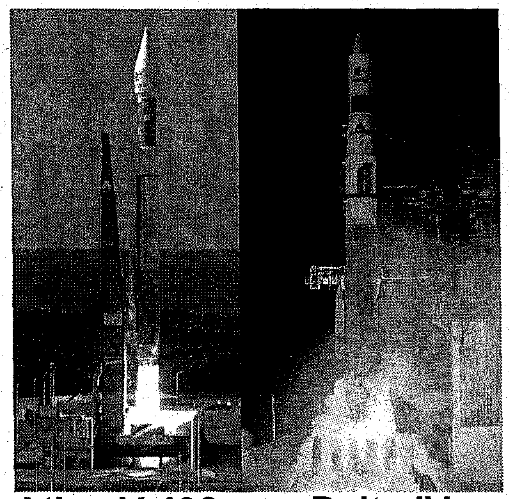
Solar power array design: Next generation Able UltraFlex structures

Baseline: Dry mass margin = 30% SEPS & 20% Chemical
SEPS propellant contingency = 10%
SEPS propulsion redundancy = 1 DCIU/PPU/Thruster

(1) 2002 NASA Aerocapture ISPT study & current study
(2) Robert Bailey, ISPT Aerocapture Systems Analysis Review, August 2002, includes science instruments
(3) Robert Bailey, ISPT Aerocapture Systems Analysis Review, August 2002, growth value of lander mass

Launch Vehicles Investigated for Analyses

Launch Mass to C_3=0 (kg)



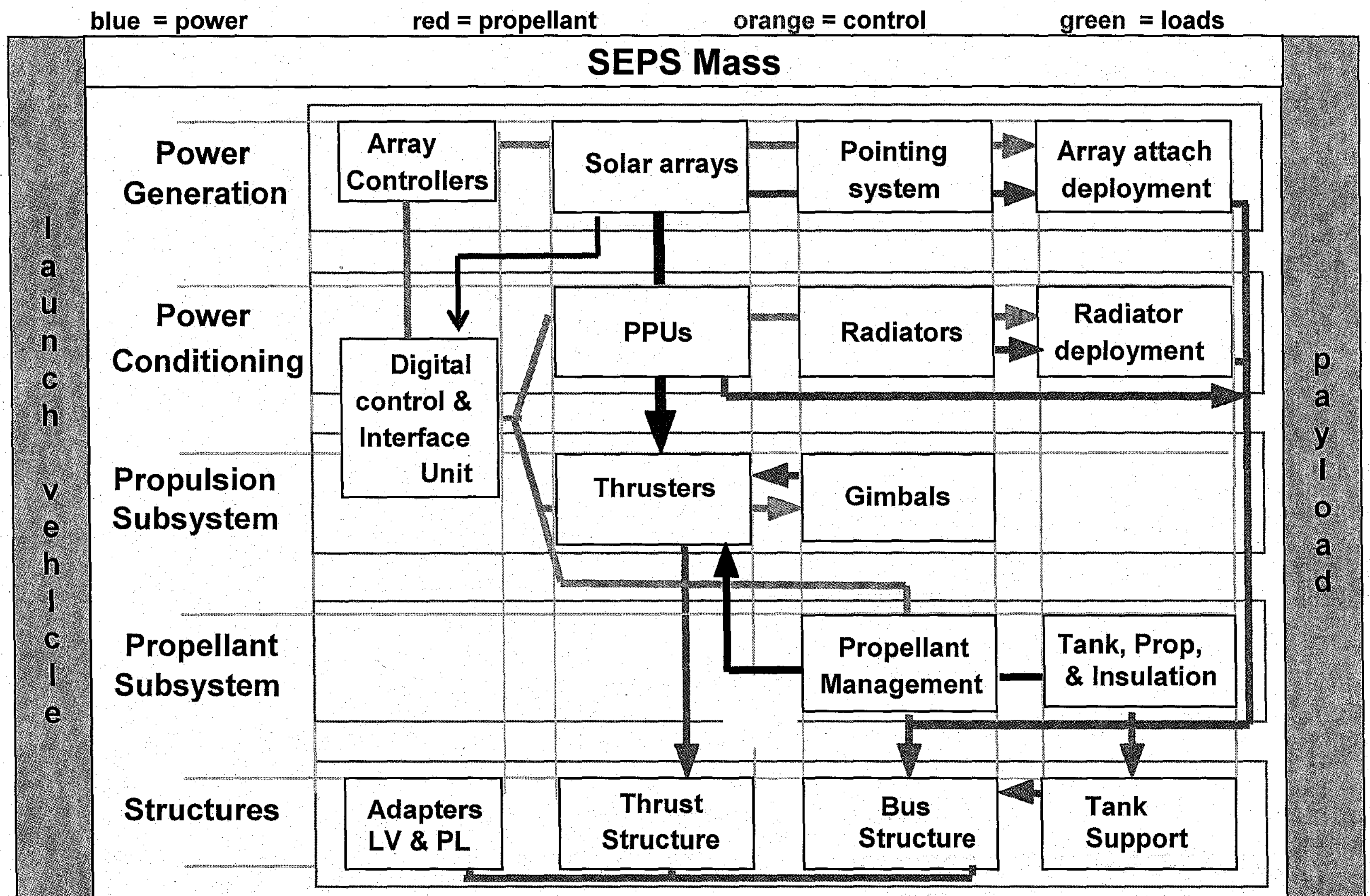
Atlas-V 400 Delta-IV

Launch Vehicle	LV Capability (0% margin)	Estimated cost[3] (millions)
Delta-IV Heavy[1]	9306	~ 140-170
Atlas-V 431	5536	~ 80 - 90
Atlas-V 421	4880	~ 75 - 80
Delta-IV 4450[2]	4583	~ 95 - 110

Wide range of launch vehicles available with modest cost differential over range of possible medium class choices

(1) Launch vehicle used for the ISPT all chemical comparison case
(2) Launch vehicle used in NASA 2002 ISPT Aerocapture study case
(3) AIAA / Isakowitz, 3rd Edition, 1999

SEPS System Model



- SEP system models are detailed and incorporate expert based algorithms
- Developed by SAIC, Huntsville Alabama, under contract to NASA's ISPT program
- Applied to technology assessment activities in support of NASA's ISPT program

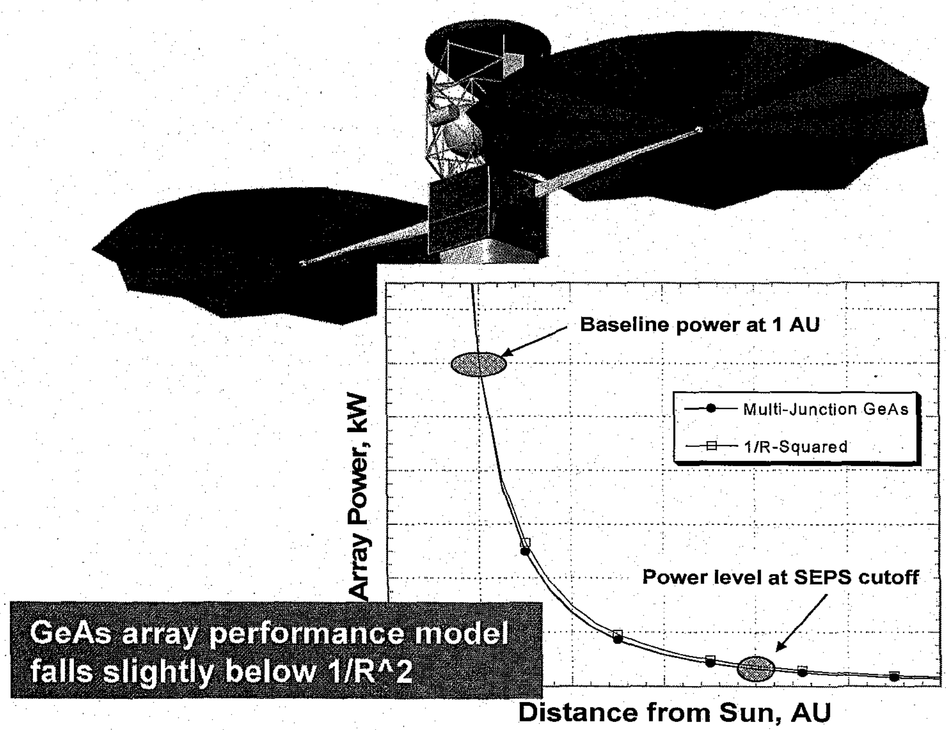
SEP System Assumptions

SEPS Baseline Power

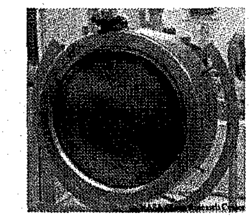
Total array power (end-of-life): 30 kWe @ 1AU

Maximum power to ion propulsion system = 25 kWe

Technology and configuration: Two 15 kWe Ultraflex Arrays, α ~ 200 w/kg



SEPS Propulsion



Thruster: NEXT, 3600 sec Isp
NEXT, 4000 sec Isp
High Isp Throttling
Max Power ~ 6.0 kWe
40 cm grids

Cases investigated:
- 4 Thrusters, 25 kWe into IPS
- 5 Thrusters, 31.25 kWe into IPS

Power Processing Unit: NGI, 6.25 kWe

Propellant management system: NGI

Propellant: Xenon

SEPS Propulsion Contingencies, Redundancy, & Other Assumptions

Baseline Contingencies:

- LV: 2% of LV nominal capacity baseline (10% also investigated)
- Propellant: 10% of total deterministic propellant
- Dry mass: 30% of dry mass
- Power: *5% added to baseline BOL power*

Redundancy:

- One extra ion system (thruster, PPU, Propellant Distribution String, & DCIU)

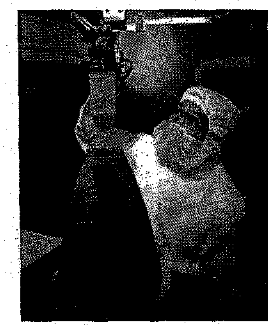
Others:

- Propulsion system duty cycle = 95% baseline
- ACS is provided by Ion Propulsion System during low thrust burn
- ACS is provided by RCS for periods when Ion Propulsion System is not active
- SEPS Housekeeping Power = 0.25 kW
- 2% of array area added to array per year of propulsion power-on time for end-of-life requirements

Conservative analysis margins and contingencies

SOA Chemical Systems: Propellant, Engine, & Tanks

- Baseline Storable; NTO/N_2H_4
  - Examples:
    - TRW ADMLAE Pressure Fed
      - Isp 330 sec
    - Aerojet R-4D-15DM
      - Isp 328 sec
  - Pressure fed
  - Off-the-shelf technology



Thrust:	102 lbf (454 N)
Propellant:	N_2O_4/N_2H_4
Pc:	100 psia
Tc:	3600 deg F
Tank Press:	250 psia
Mix Ratio:	1.06
Area Ratio:	204:1
Mass:	10.6 lb (4.8 kg)
Chamber:	Iridium lined
Nozzle:	rhenium/Niobium
Length/Dia:	26.9 / 11.8 in

Aerojet R-4D-15DM pressure fed NTO/N_2H_4

- Tank Liner Thickness and Composite Overwrap Stress Factor
  - Baseline liner thickness 30 mil
  - Nominal tank stress factor is scaled from AXAF tanks

Chemical Stage Contingencies & Other Assumptions

Contingencies:

- Propellant: 3.0% of total deterministic propellant
- Dry Mass: 20% of total mass (non payload) delivered
- Power: Provided by RTG (Radioisotope Thermal Generator) & batteries
- Structure: Size based on historical data for actual spacecraft

Others:

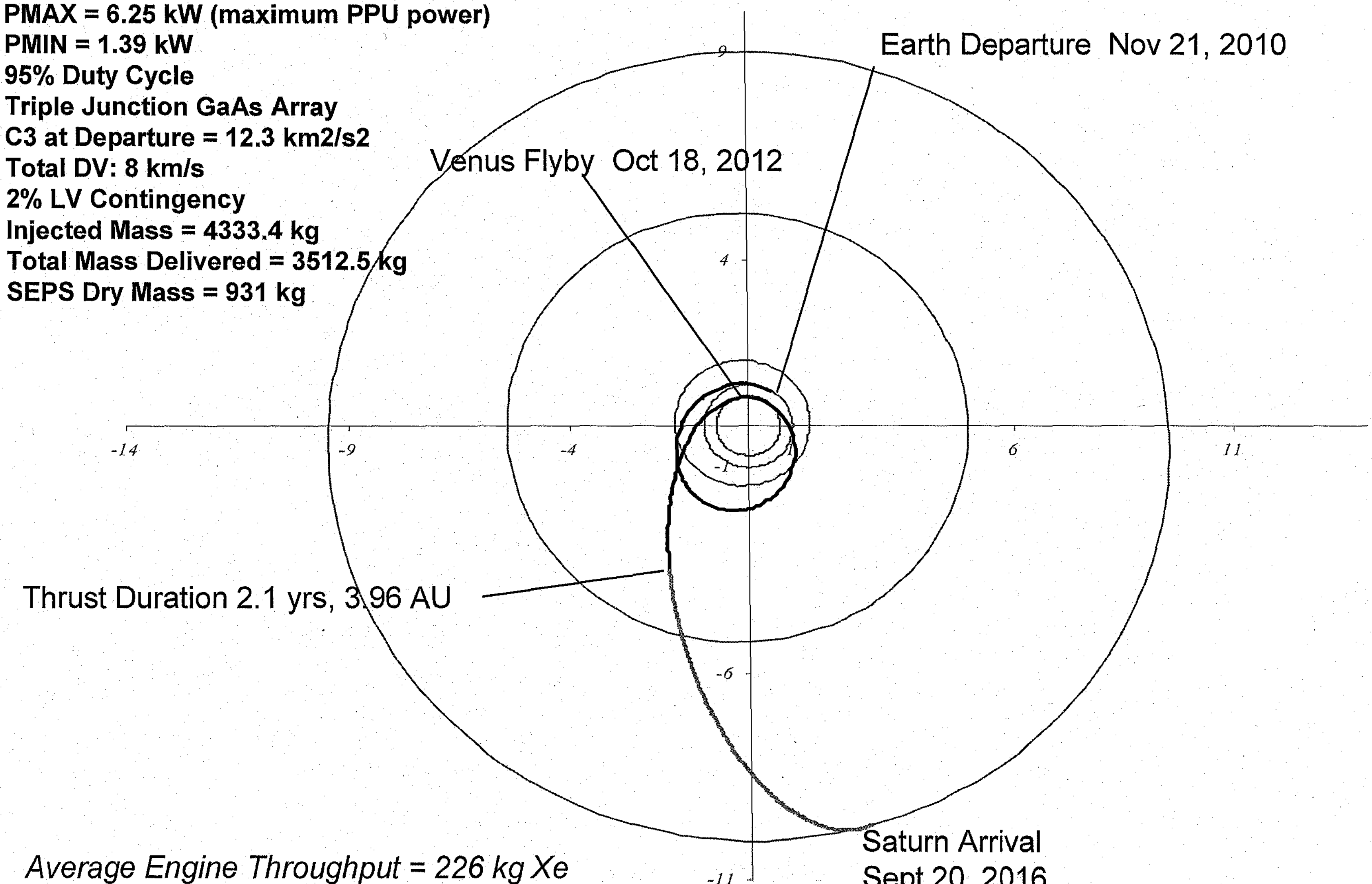
- 20 m/s total RCS ΔV, includes maintaining proper final parking orbit
- 20 m/s ΔV for course correction and aimpoint maneuvers

Conservative analysis margins and contingencies

SEPS Saturn Trajectory Sample

8.5 Year, Atlas-V 431

4 NEXT Engines @ 4000 sec Isp
Power = 25 kW into ion propulsion system
PMAX = 6.25 kW (maximum PPU power)
PMIN = 1.39 kW
95% Duty Cycle
Triple Junction GaAs Array
C3 at Departure = 12.3 km2/s2
Total DV: 8 km/s
2% LV Contingency
Injected Mass = 4333.4 kg
Total Mass Delivered = 3512.5 kg
SEPS Dry Mass = 931 kg

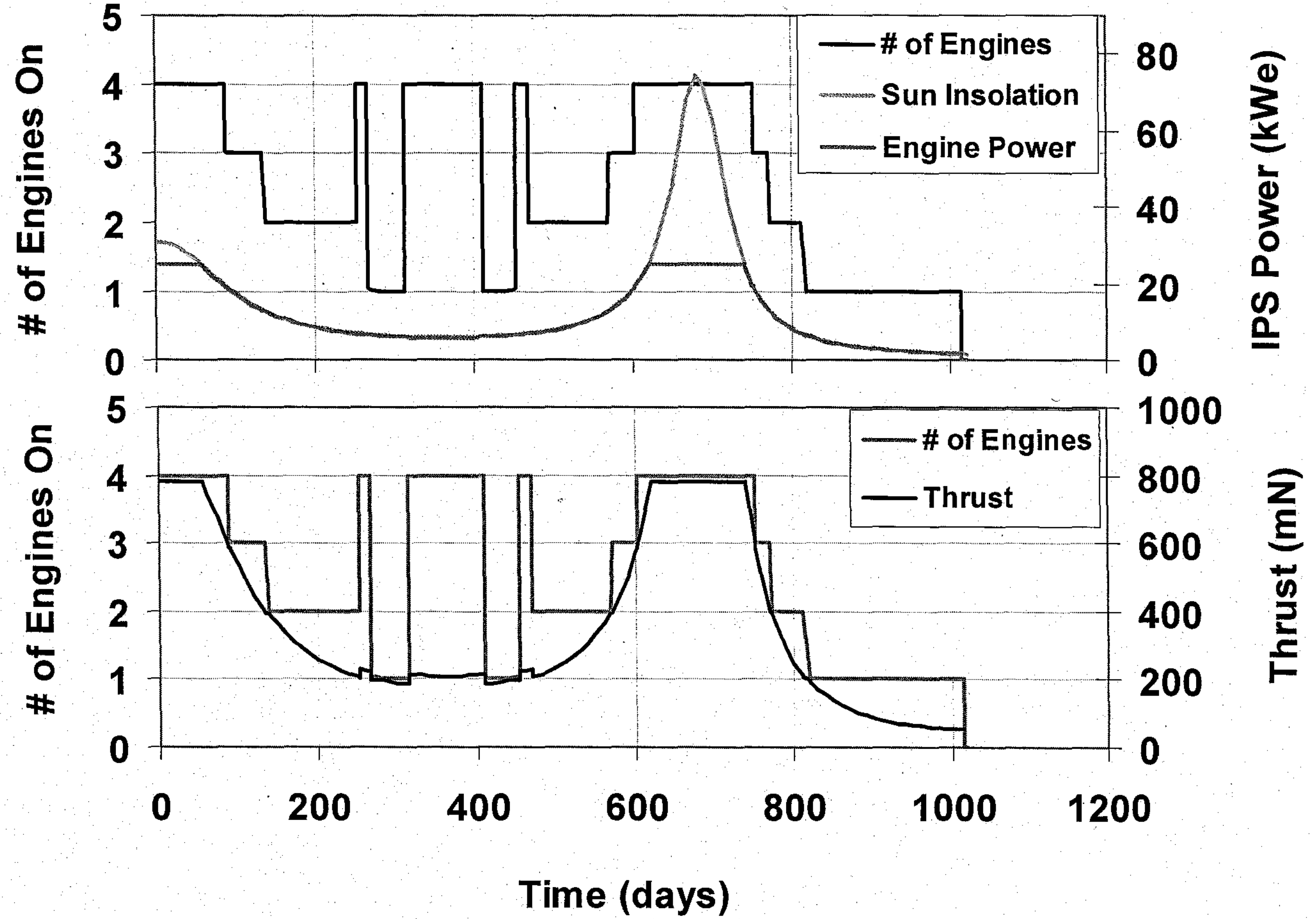


Average Engine Throughput = 226 kg Xe

Chemical capture type trajectory

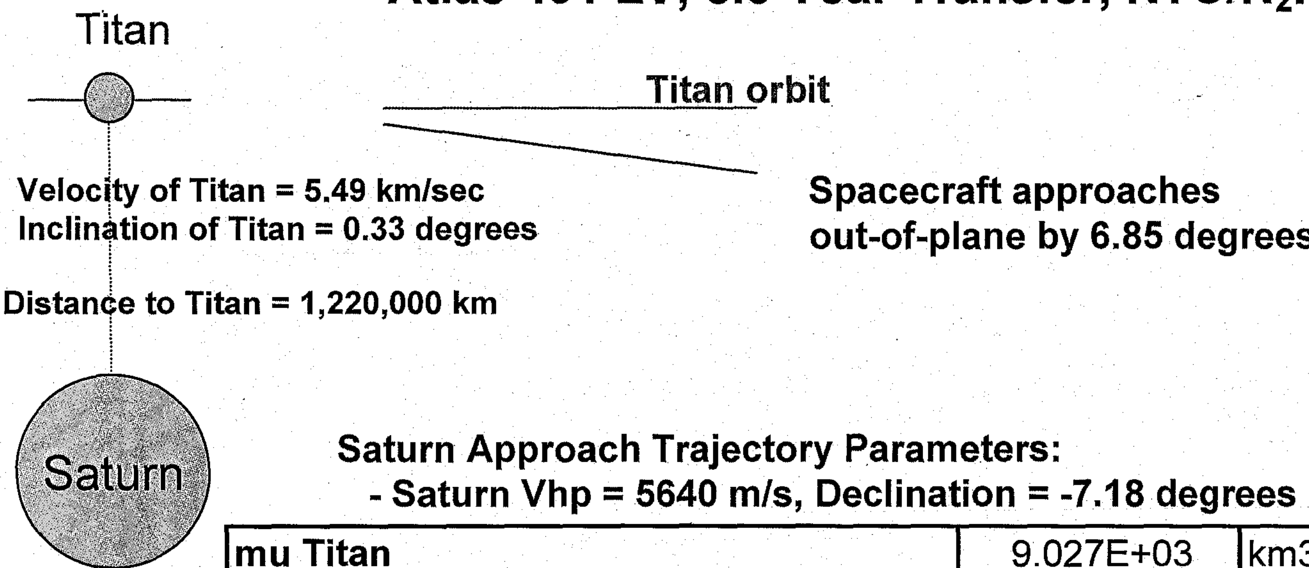
SEPS Engine/Power Profiles

High Isp Throttling, 8.5 Year, Atlas-V 431



Baseline Propulsive Capture Methodology Example

Atlas 431 LV, 8.5 Year Transfer, NTO/N_2H_4

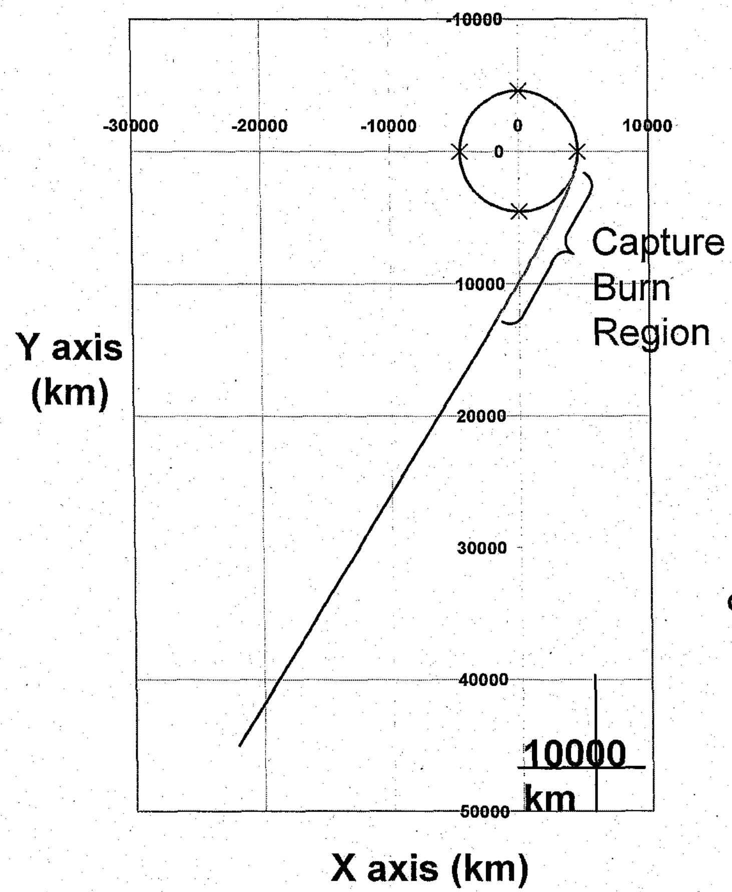


Saturn Approach Trajectory Parameters:
- **Saturn Vhp = 5640 m/s, Declination = -7.18 degrees**

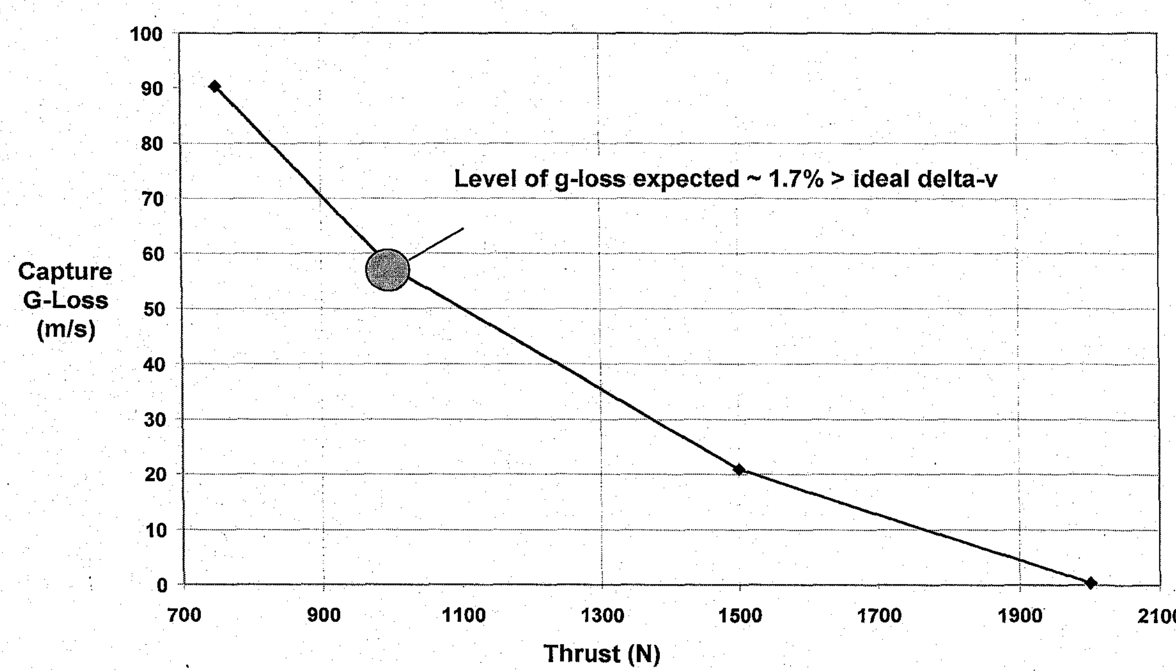
mu Titan	9.027E+03	km3/s2	
radius Titan	2575	km	
capture orbit altitude	2000	km circular	
Vhp relative to Titan	4250	m/sec	
hyperbolic velocity at Titan periapsis	4691	m/sec	
circular velocity at Titan capture orbit	1405	m/sec	
ideal capture delta velocity	3287	m/sec	
contingency: orbit insertion uncertainty	66	m/sec	2 % of ideal delta-v
g-losses	82	m/sec	2.5 % of ideal delta-v
Nav & trajectory correction delta-v	20	m/sec	
Total delta-v	3455	m/sec	

Titan out-of-plane capture modeled and accounted for in analyses

Titan Capture g-loss Analysis

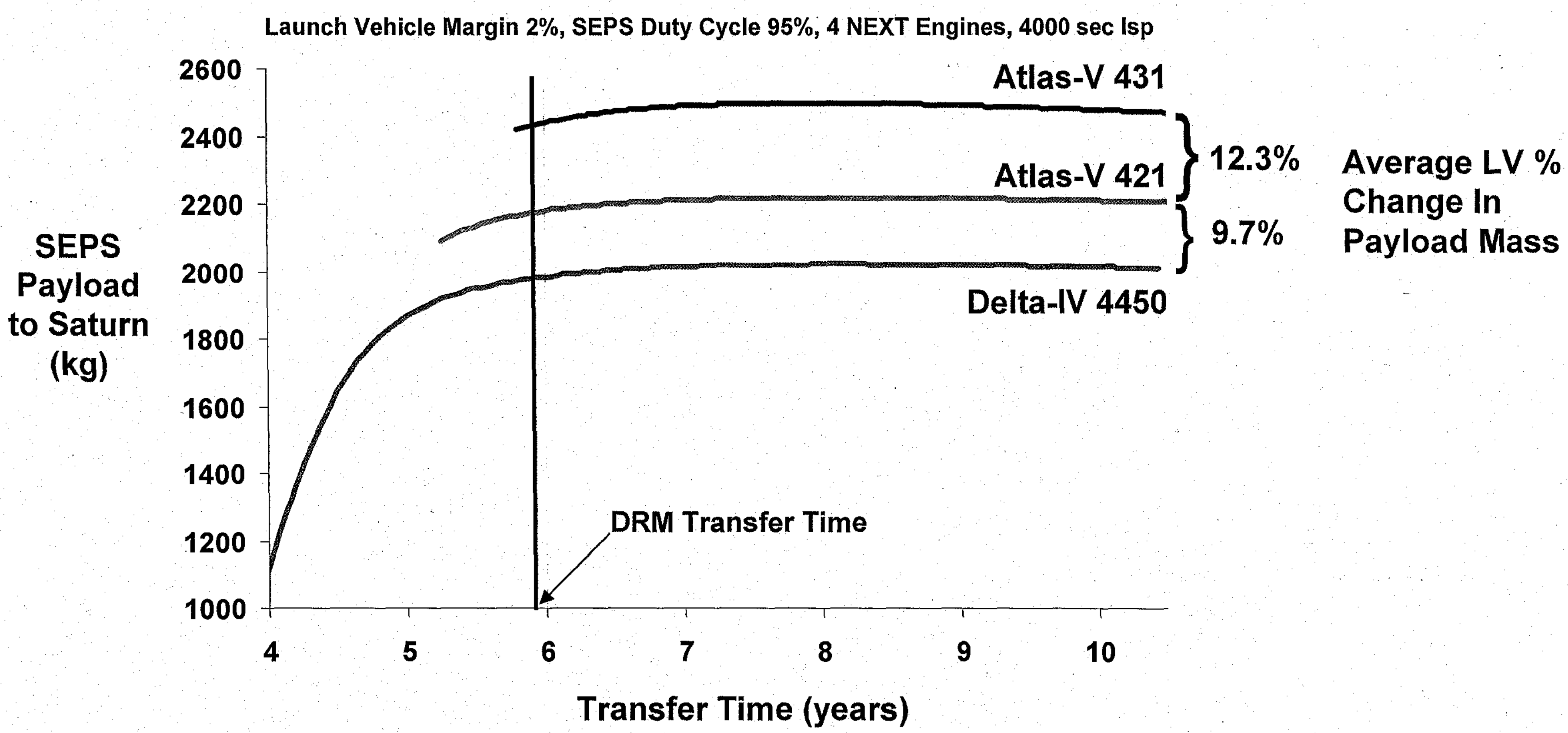


- Altitude Periapsis= 1995.5 km
- Altitude Apoapsis= 2003.3 km
- Thrust = 1000 N (two 100 lb engines)
- Burn Time = 4210.5 sec
- ΔV (m/s), ideal = 3288 ; actual = 3345.3
- Propellant mass (kg), ideal = 1287.8, actual = 1300.6
- Propellant Difference = 12.8 kg
- Gravity Loss = 57.3 m/sec



SEPS Payload to Saturn vs. Transfer Time

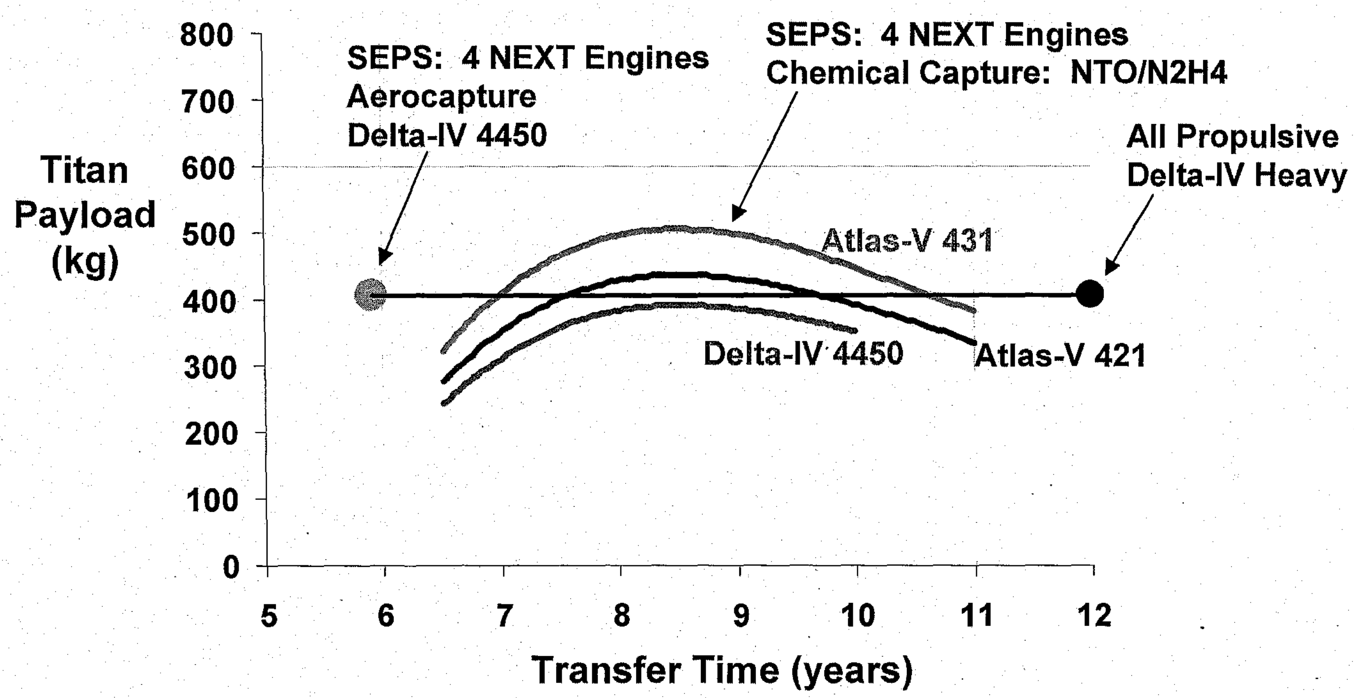
Launch Vehicle Variation



Launch vehicles investigated cover a SEPS payload range between ~1100 to ~2500 kg

Titan Payload vs. Transfer Time

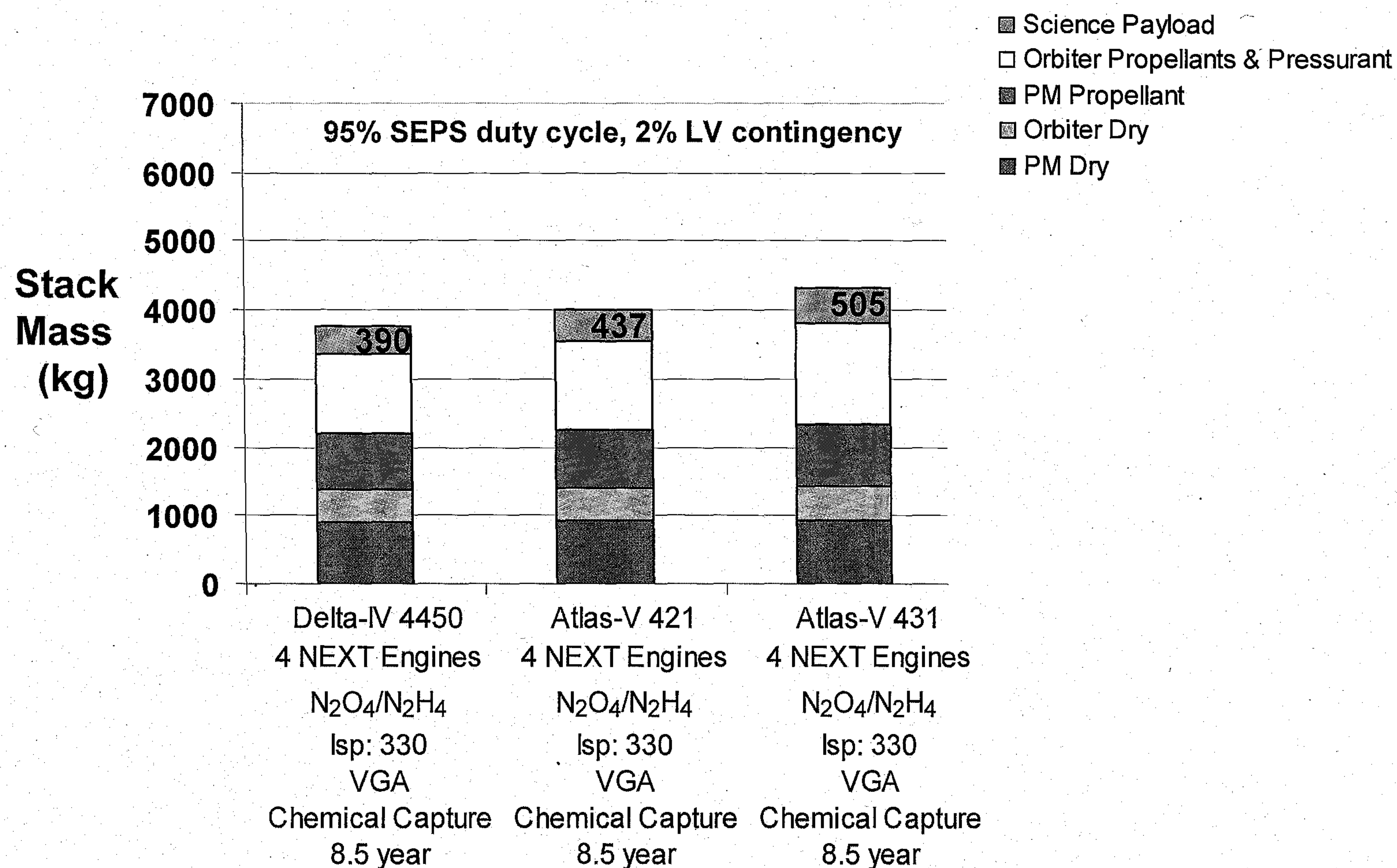
Baseline NTO/N_2H_4



For SOA chemical, a minimum LV capacity of Atlas-V 421 is required to accomplish the mission

Stack Mass

Launch Vehicle Comparison, 4 Ion engines @ 4000 Isp, NTO/N_2H_4



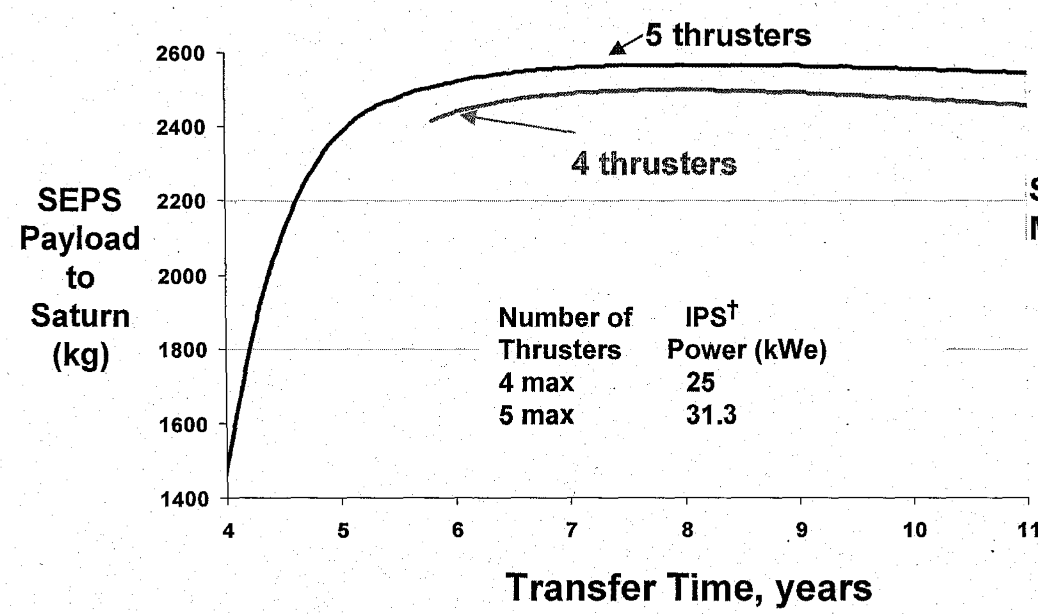
For SOA storable, the mission could be accomplished with the Atlas 421 or Atlas 431.

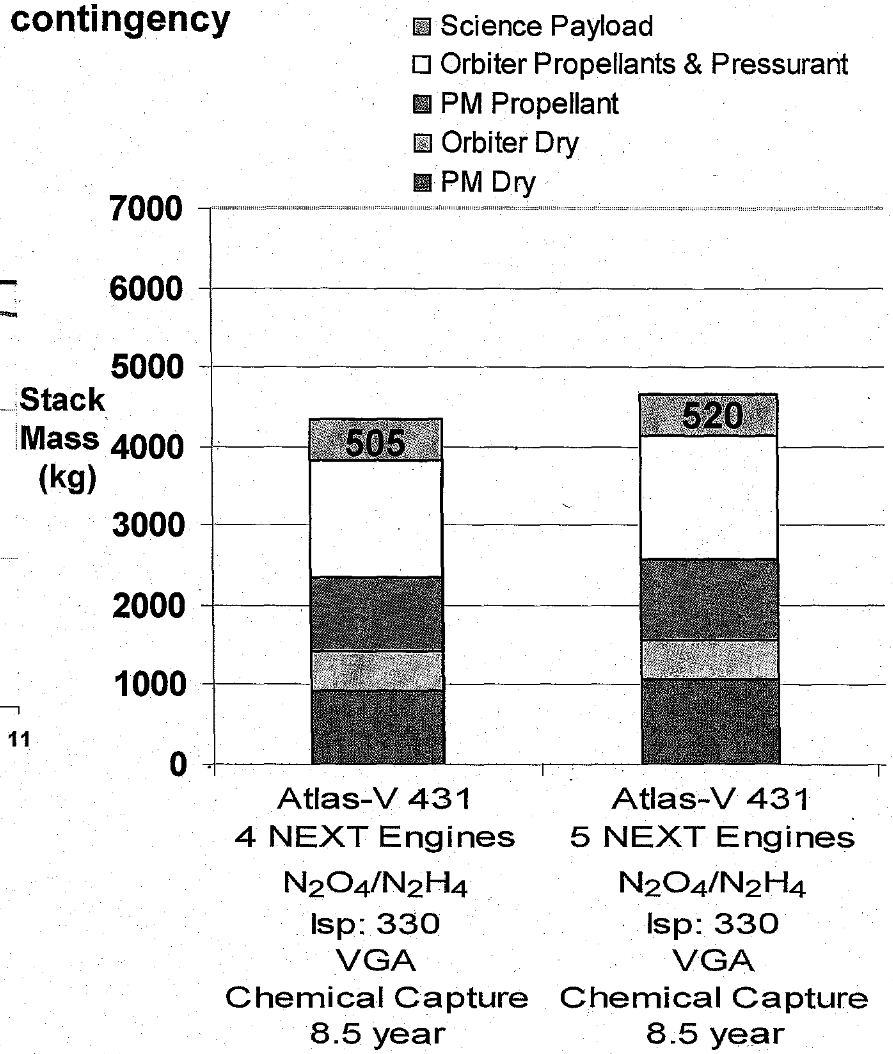
(1) MSFC PRC Seminar, Oct 29, 2002

SEPS Payload vs Transfer Time

Variation in Number of Operational Thrusters

LV: Atlas 431, 2% contingency





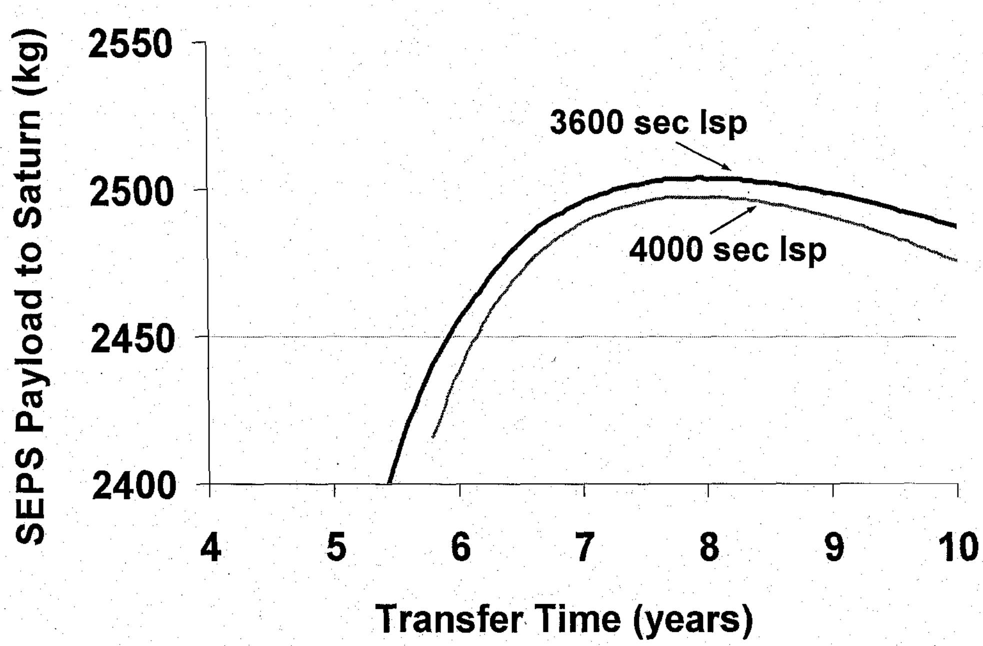
• 5 thruster SEP configuration delivers more mass to Saturn than the 4 thruster configuration

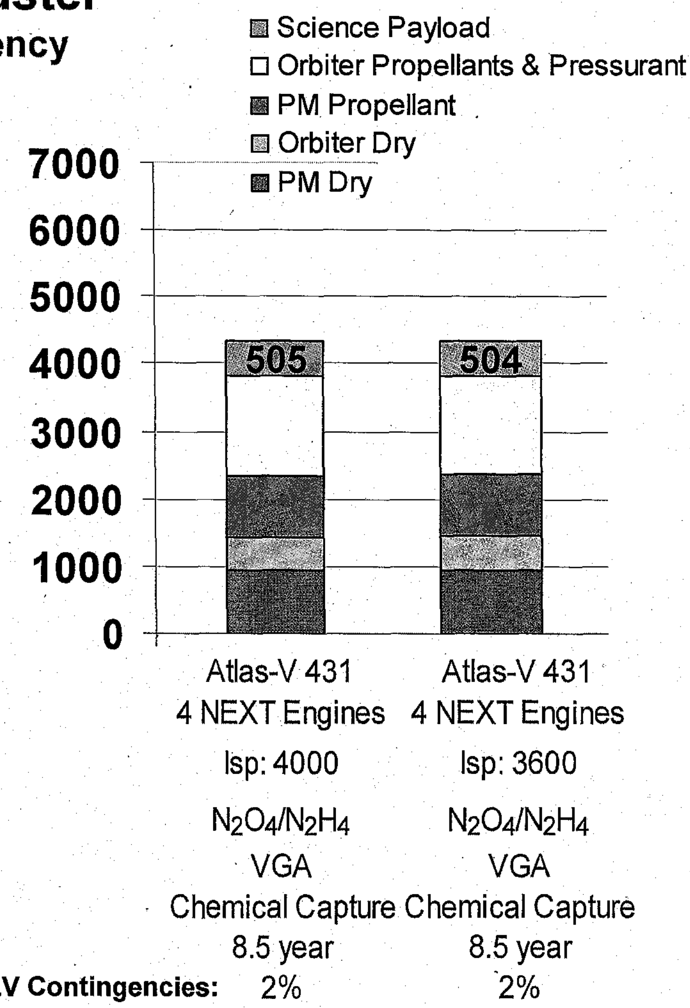
† Nominal maximum power into a PPU = 6.25 kWe

SEPS Payload vs Transfer Time

Variation in Isp of Thruster

LV: Atlas 431, 2% contingency





- 5 thruster SEP configuration delivers more mass to Saturn than the 4 thruster configuration

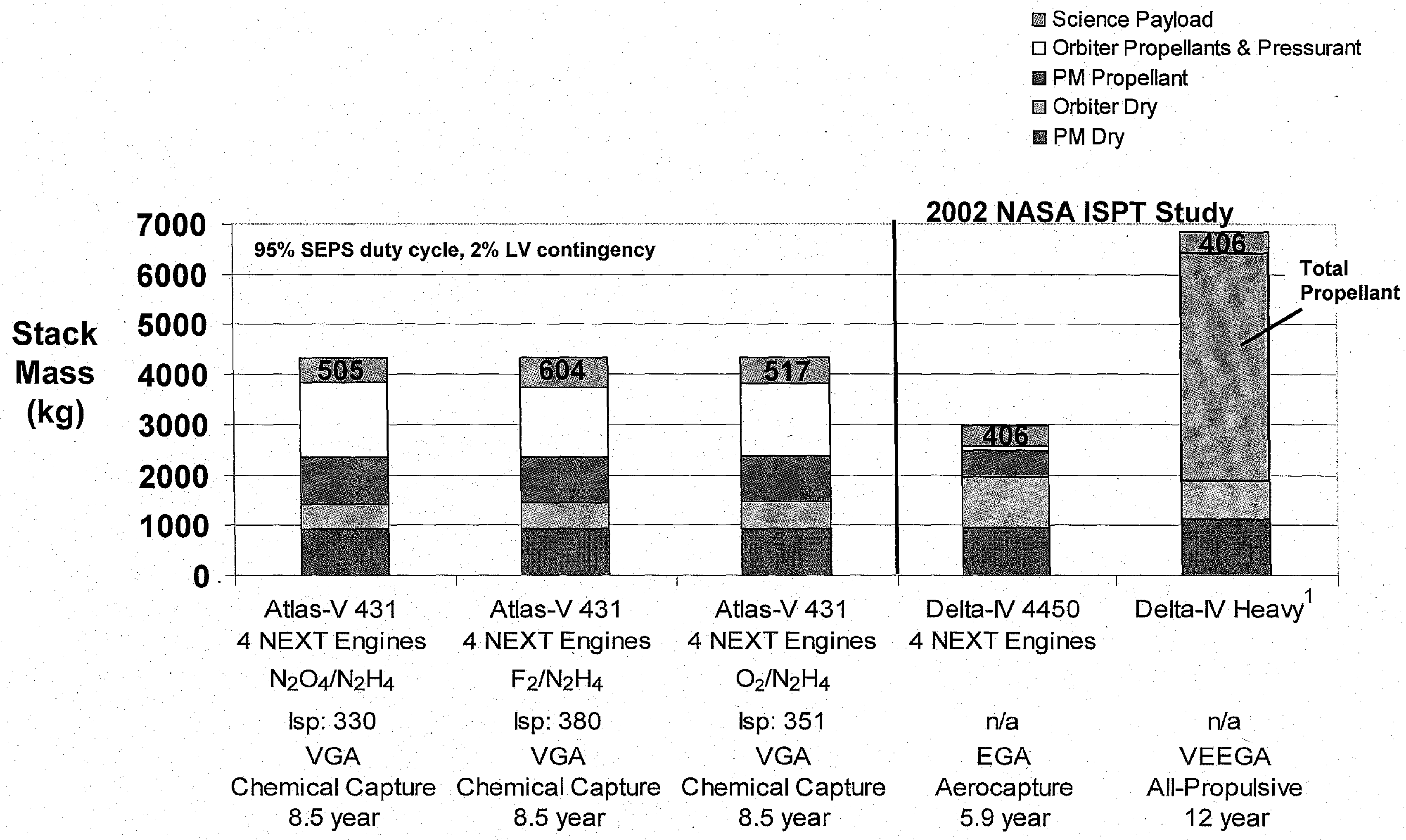
Advanced Technology Parametric Study

Advanced Chemical Systems: Propellants, Engines, & Tanks

- Option 1: Advanced Storable; LOX/N_2H_4
  - 351 Isp
  - Pressure fed, 100 psia chamber pressure
  - Requires active refrigeration of LOX
  - Experimental
- Option 2: Advanced Storable; F_2/N_2H_4
  - 380 Isp
  - Pressure fed, 100 psia chamber pressure
  - Very reactive and toxic
    - Fluorine oxidizers react with their containment vessels; internal surfaces must be passivated.
    - requires active refrigeration of LOX
  - Experimental
- Option 3: Advanced Storable; Monopropellant
  - 275 Isp
  - Pressure fed, 100 psia chamber pressure
- Tank Liner Thickness and Composite Overwrap Stress Factor
  - Liner thickness 5 mil to 30 mil
  - Stress factor from 70% to 130% of nominal
  - Nominal tank stress factor is scaled from AXAF tanks

Stack Mass

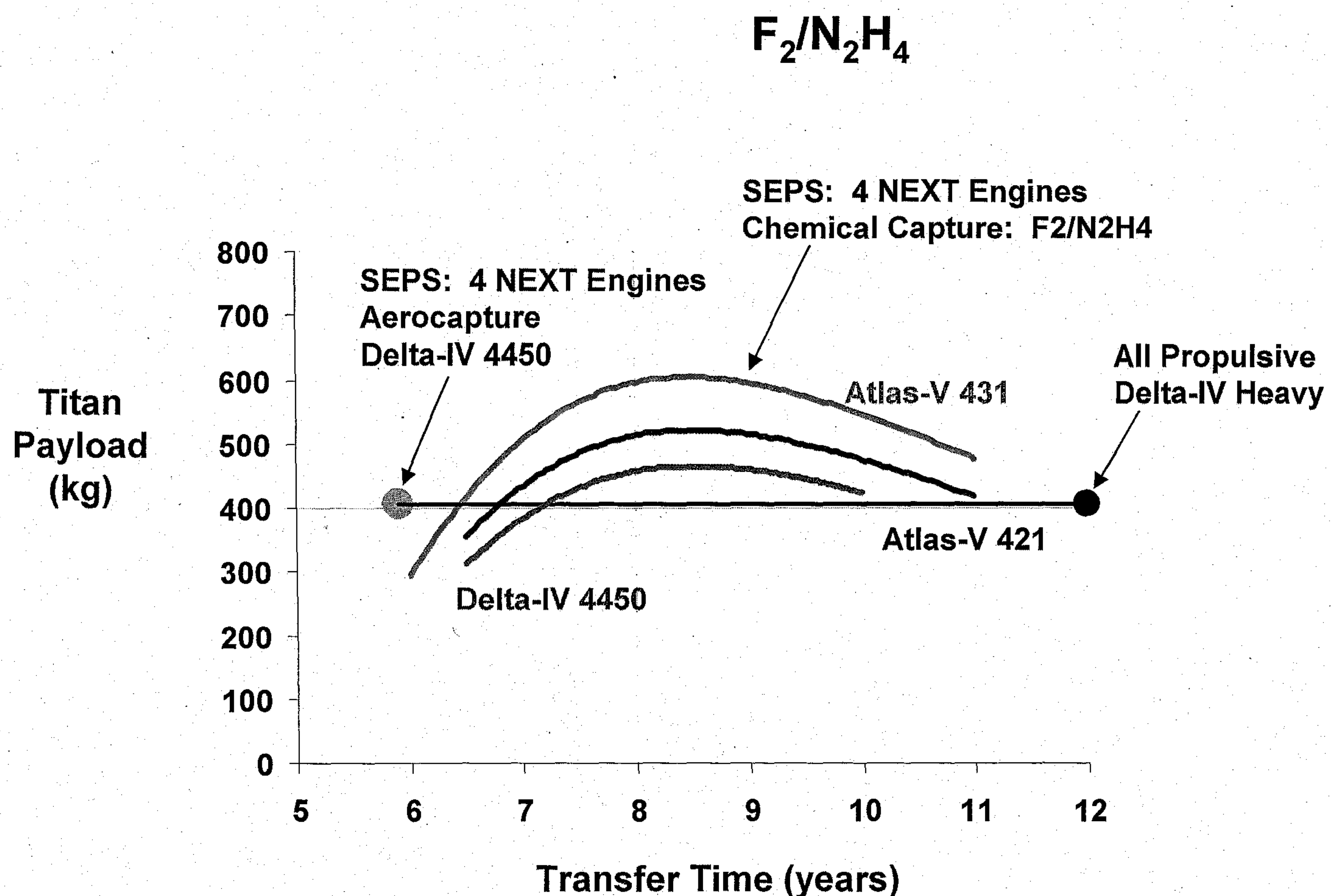
Chemical Technology Comparison



For propellants investigated and for an Atlas 431 LV, significant margin over reference payload of 406 kg can be realized

(1) MSFC PRC Seminar, Oct 29, 2002

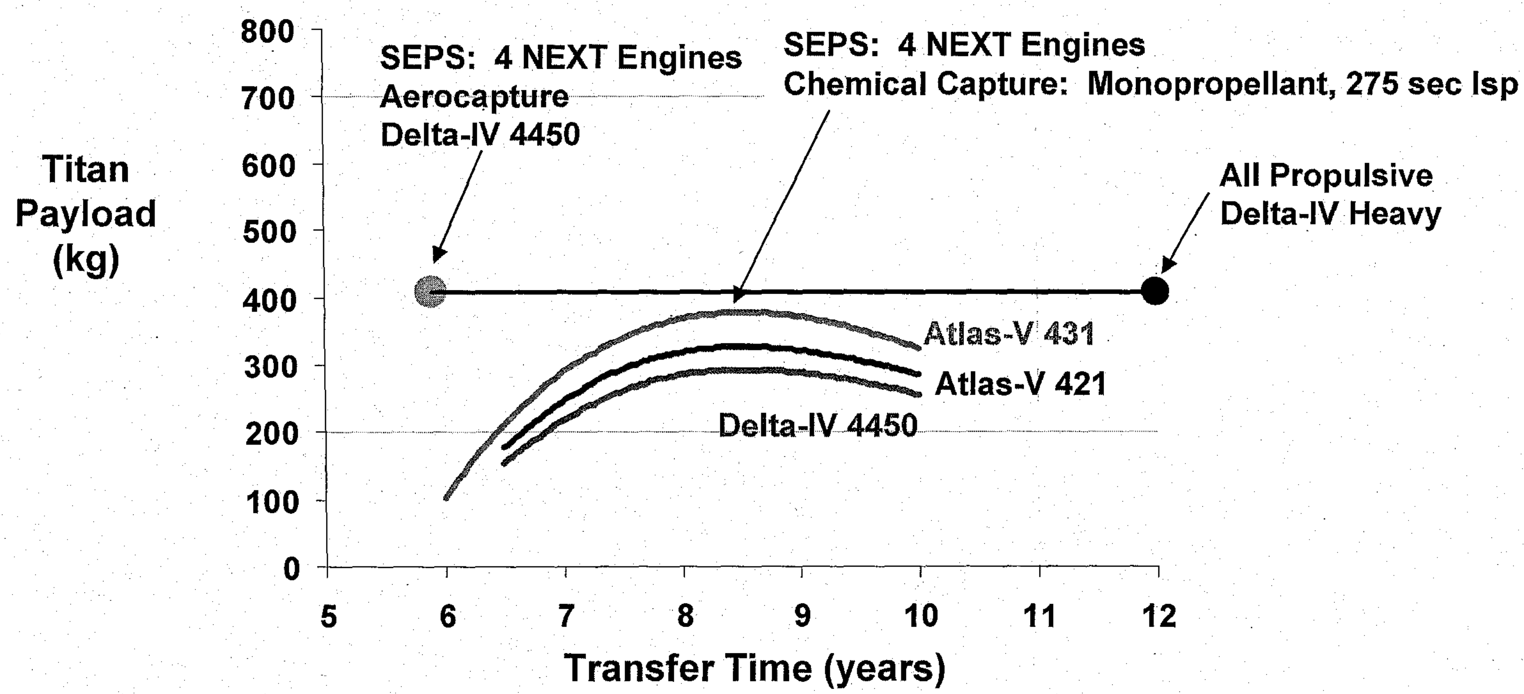
Titan Payload vs. Transfer Time



- For F_2/N_2H_4, a minimum LV capacity of Delta-IV 4450 can accomplish the mission
- F_2/N_2H_4 provides significantly more performance capability than SOA chemical
- Transfer time for the Atlas-V 431 case is less than 6.5 years

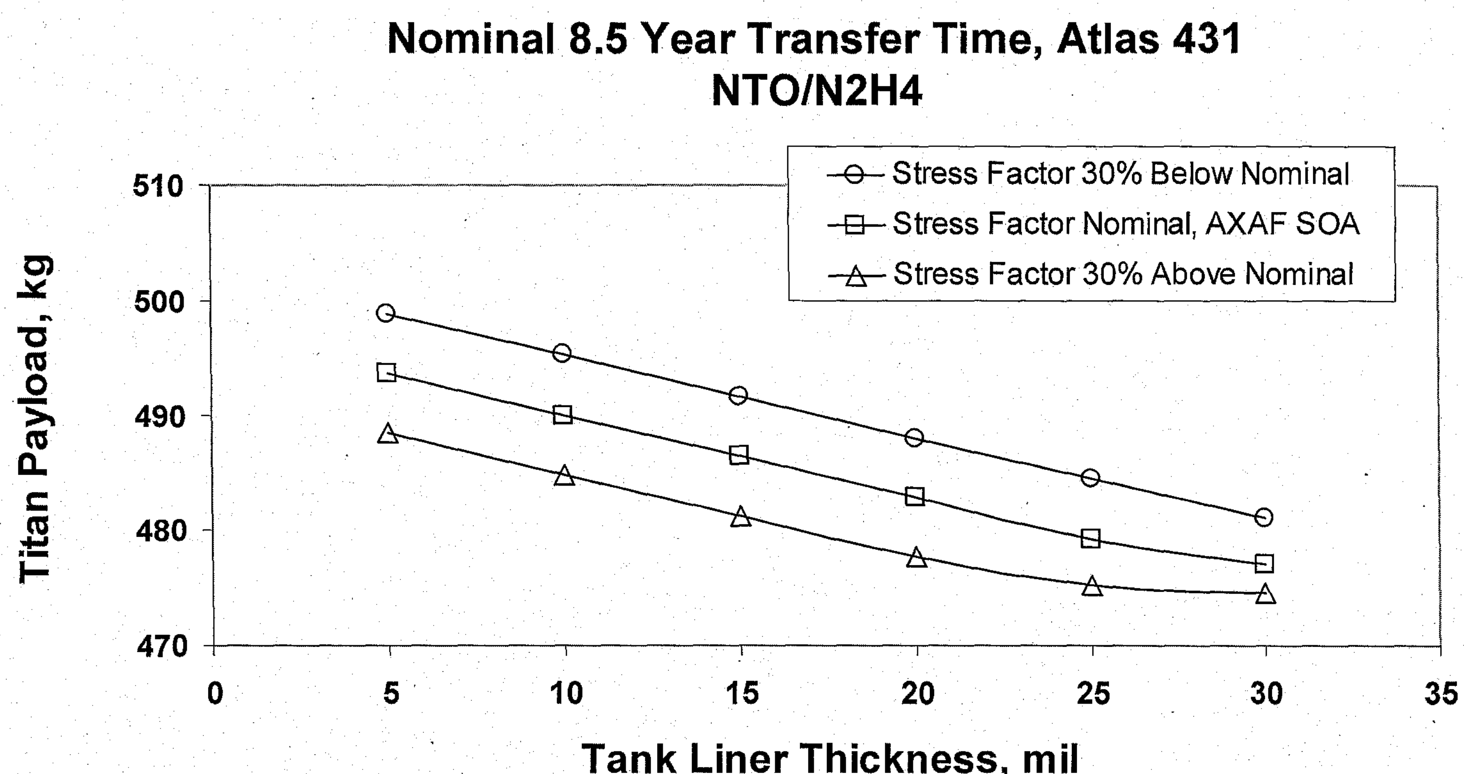
Titan Payload vs. Transfer Time

Advanced Monopropellant, 275 sec Isp



- For an advanced monopropellant of 275 sec Isp, a medium LV cannot accomplish the mission

Payload Dependence on Variation in Tank Stress Factor



Varying tank liner thickness from SOA to 5 mil and reducing required stress factor by 30% from SOA increased payload over the baseline (43 kg) by 50%.

Advanced Propulsion Analyses Conclusions

- NTO/N_2H_4 Propulsion
  - Atlas V 421 is the minimum launch vehicle to deliver the study payload: 67% orbiter payload margin.
  - Atlas V 431 can deliver ISPT study payload with a 215% orbiter payload margin.
- F_2/N_2H_4 Propulsion
  - Provides significant performance increase over NTO/N_2H_4 (Atlas 431 LV) : 100 kg of orbiter payload
- LOX/N_2H_4 Propulsion
  - Provides an improvement over NTO/N_2H_4 (Atlas 431 LV) of only 12 kg of orbiter payload: probably does not warrant a new engine development program for 100 lbf class engines
- Monopropellants
  - Provides no improvements over SOA chemical unless the Isp is significantly greater than 275 sec
- Tank Technologies
  - decreasing tank liner thickness and reducing pressure factor can provide a significant increase in payload delivery capability over SOA
- Results indicate that SEP/Chemical systems can accomplish the mission using "conventional" chemical technology:
  - With longer trip times and larger launch vehicles required than the SEP/AC option
  - In much shorter trip times and with smaller launch vehicles than an all-propulsive chemical mission
  - With larger overall stack mass required than SEP/AC option